AGARD-R-716-ADDENDUM

AGARD-R-716-ADDENDUM

AD-A148 800

AGARD

ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

7 RUE ANCELLE, 92200 NEUILLY SUR SEINE, FRANCE

AGARD REPORT No.716

Composite Structure Repair

Addendum

This document has been approved for public release and sale, its distribution is unlimited.

DTIC ELECTE DEC 2 8 1984 A

NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION AND AVAILABILITY ON BACK COVER

DTIC FILE COPY

20000803204

84 12 14 016

AGARD-R-716-Addendum

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

ADDENDUM TO

AGARD Report No.716

COMPOSITE STRUCTURE REPAIR

Dist A-1

Papers presented at the 58th Meeting of the Structures and Materials Panel,
in Sienna, Italy, on 2—6 April 1984

THE MISSION OF AGARD

The mission of AGARD is to bring together the leading personalities of the NATO nations in the fields of science and technology relating to aerospace for the following purposes:

- Exchanging of scientific and technical information;
- Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;
- Improving the co-operation among member nations in aerospace research and development;
- Providing scientific and technical advice and assistance to the North Atlantic Military Committee in the field of aerospace research and development;
- Rendering scientific and technical assistance, as requested, to other NATO bodies and to member nations in connection with research and development problems in the aerospace field;
- Providing assistance to member nations for the purpose of increasing their scientific and technical potential;
- Recommending effective ways for the member nations to use their research and development capabilities for the common benefit of the NATO community.

The highest authority within AGARD is the National Delegates Board consisting of officially appointed senior representatives from each member nation. The mission of AGARD is carried out through the Panels which are composed of experts appointed by the National Delegates, the Consultant and Exchange Programme and the Aerospace Applications Studies Programme. The results of AGARD work are reported to the member nations and the NATO Authorities through the AGARD series of publications of which this is one.

Participation in AGARD activities is by invitation only and is normally limited to citizens of the NATO nations.

The content of this publication has been reproduced
directly from material supplied by AGARD or the authors.

Published August 1984

Copyright © AGARD 1984
All Rights Reserved

ISBN 92-835-1476-9

Printed by Specialised Printing Services Limited
40 Chigwell Lane, Loughton, Essex IG10 3TZ

CONTENTS

REPAIR OF COMPOSITES

T.M. Donnellan, E.L. Rosenzweig, R.E. Trabocco
Naval Air Development Center
Warminster, Pa. USA

J.G. Williams
Aeronautical Research Laboratory
Melbourne, Vict., Australia

SUMMARY

This paper discusses developments in composite repair technology. The planned use of composites in primary aircraft structure has necessitated the development of a specialized repair technology. The types of damage seen in composites differs from conventional materials. Delamination is the most serious composite defect since it can exist without causing surface damage. The significance of a particular size damage will depend on the location in the structure and structural requirements of the component. The two techniques used to repair composites are bonded and bolted approaches. The approach selected for a particular application depends on the type of component, amount of strength restoration desired, aerodynamic considerations, and the materials and equipment available at the repair site. Composite damage in service has been limited since very few structural components are in use. A number of programs have been performed which have evaluated repair techniques under simulated operating conditions. Programs are described in which bonded and bolted repair techniques were validated. Generic repair development programs are described in which the repairs were performed with equipment and techniques possible in the service environment. The results of these studies showed that the repairs restored the component structural integrity. Repair programs on specific military and commercial aircraft are also described. The components studied included an AHI Rotor Blade, a S-3A Spoiler and a L-1011 Fin.

INTRODUCTION

Composite materials are being utilized for commercial and military applications in an increasing fashion.[1,2] Their combination of superior mechanical properties, design versatility and low density make them attractive structural candidates.

The utilization of composite components on aircraft has necessitated the development of composite repair technology.[3] Composite repair programs have been undertaken both by commercial and military sources.[4,5] Repair concepts have been developed which include bolted patch repairs and bonded patch repairs. The particular technique used to repair a given component will be dependent on the specific capabilities available and the requirements for structural integrity.

This paper will summarize developments in composite repair technology. The use of bolted and bonded repair procedures in the commercial and military areas of operation will be discussed.

COMPOSITE DAMAGE

Many types of damage are possible in composite structures. Defects may be introduced during composite manufacture due to errors in manufacturing procedure. Damage may be initiated during the life of the component as a result of exposure to the service environment, by accumulation of minor damage sustained in the normal use of the composite, from occasional exposure to an abnormal environment, or from a local mechanical overload due to misuse. The damage may affect the component in two simple ways. First, the defect may degrade the strength or stiffness of a component to a level below its design limits. A defect of this nature must be detected and the component repaired or replaced. Of more concern, are smaller defects which do not immediately degrade mechanical performance, but which will grow under service conditions to a point where strength or stiffness is unacceptable.

Most of the service damage experienced with composite structures is due to impact with a foreign object or mishandling. The damage resulting from an impact can range from small surface dents to internal delaminations and through penetration. Because of the tendency for delaminations to form, it is necessary to inspect the damaged area for such occurrences. One of the most important aspects of a repair action is to establish the area which is defective. Ultrasonics and x-radiography are the most commonly utilized inspection techniques. Such inspection provides a means of determining damage significance and repairability for a specific component.

The size limitation of repairability is dependent on the type of component construction. For monolithic skin or skin stringer constructions in which composite thickness ranges from .1 in. to .4 in., repairable damage is currently restricted to a 4 in. diameter area. Another type of structure is composite honeycomb in which composite face sheets which may range in thickness from .03 in. to .16 in., are bonded to aluminum or organic core materials. In this case, the limit on repairable delaminations has been set at 12 in^2 and the limit on through penetration has been set at approximately a 6 in. diameter damage.

REPAIR CONCEPTS

The objective of a repair action is to restore structural integrity to a damaged component. The particular procedure used will depend on the type of component, amount of joint efficiency required and aerodynamic considerations.

The repair environment will also affect the technique used in repair operations. At a rework facility, major repairs are possible, but at a forward or field position the limits on repairability are much more restricted. The rework facility is equipped with freezer storage for materials, autoclave capability for processing, and tools designed for repair applications. Also, the level of expertise available in a rework facility is higher than that available in the field. Thus, parts which are easily detachable from the aircraft can be repaired under ideal autoclave conditions. Manufacturing materials, both pre-pregs and adhesives, can be used in the repair procedure. The rework facility is capable of performing all repairs required for the aircraft.

The field repair environment is more severe. No freezer storage may be available, and the materials used should be capable of ambient temperature storage. There is no access to autoclave facilities and thus repairs must be performed with heating blankets. Compaction and conformational pressure is usually limited to vacuum bag pressure of 1 atm. The personnel and tooling available restrict the repairs to external bonded patches and simple bolted patches. The two basic repair concepts are bonded and bolted patch techniques. Each has application in specific repair situations.

Bonded Repair Techniques

Bonded techniques can be used in situations which range from cosmetic to primary structural repairs. Cosmetic repairs refer to damage which is not structurally significant (i.e., dents, missing surface plies). The repair is made to restore aerodynamic smoothness. In these repairs, a potting compound or a liquid adhesive is spread into the damaged area and formed to the component's contour.

Injection repairs are another type of repair procedure which are used for minor disbonds or delaminations. In this procedure a number of holes are drilled to the depth of the damage. Filler resin is heated to decrease viscosity and injected under pressure until the excess flows out of adjacent holes. Pressure can be applied to the repaired area to insure mating of adjacent regions. If serious damage is encountered, then more rigorous repair procedures must be employed. There are two types of bonded patches which can be used to repair structural damage, namely, flush scarf patches and external patches.

Flush scarf type bonded repairs are used where aerodynamic smoothness is essential. This approach provides the highest joint efficiency (60-100%) of any repair technique. Scarf repairs have been used on critical components where load concentration and eccentricities must be avoided. Thick monolithic structures lend themselves to such a repair since an external patch would cause excessive out of moldline thickness and unacceptably high bondline peel and shear stresses.

The flush repair procedure requires careful preparation of the damaged area to obtain the correct scarf angle and dimensional tolerances. The laminate orientation of the patch must match that of the damaged section which has been removed. Scarf patches can be cocured (cured on the damaged laminate) or precured (cured and then secondarily bonded onto the damage) and they can either be single or double scarfs (figure 1). The double scarf permits a shorter joint length and requires less material be removed from the substrate.

The external patch technique is less critical in nature than a scarf approach and is used typically on thinner laminates. External patch repairs require less preparation than scarf repairs. Limited back side access or substructure interference in a damaged area would favor the use of an external patch repair.

In this approach the load is taken over and around the damaged area. The bending strains due to the eccentric load path must be considered in the patch design. The patch must also be capable of withstanding the high peel and shear stresses which develop at the edge of the damage. In order to minimize this effect and "fair in" loads to the substrate, the patch plies are stepped in diameter, usually with the largest ply on the bottom of the stack and decreasing in size to the top. An external patch repair could be expected to restore 60-80% of component strength. The simple nature of this approach lends itself to remote or field maintenance scenarios.

The materials used in bonded repairs must be capable of operating within the thermal environment of the damaged component. For most aerospace applications this necessitates the use of elevated temperature curing adhesives and composites. A typical processing schedule would include both heat to cure the polymer and pressure for ply compaction and conformation of bonding surfaces. In certain instances, ambient temperature cure systems may be sufficient.

Moisture presents a problem for cured epoxy composites since it acts as a plasticizer and reduces elevated temperature matrix dominant properties. It presents an even more significant problem for bonded repairs which require heating above 212°F for resin curing. When elevated temperature curing is performed in the presence of moisture, either in the adhesive or laminating resin of the composite substrate, voiding frequently results. The

most deleterious form is bondline voiding since bond strength can be greatly decreased. Heating the damaged area prior to cure has been used to drive off excessive moisture. However, this approach is time consuming and relatively impractical.

Bolted Repair Techniques

Bolted metal patch composite repair is an alternate approach to the bonded repair concept. Bolted repairs can be used in cases where bonded patch repair of a thick laminate may result in shear stresses beyond the limit of the adhesive strength. This method is also appropriate when a bonded scarf approach would be too complex in terms of preparation and material removal.

Bolted repairs can be used when the laminate thickness exceeds .03 in. Thinner laminates cannot withstand the bearing loads induced by the bolts. Bolted patches can be applied from one side or from two sides with a backing plate. These backing plates can be inserted through the damage to gain access to the back side. If the plates are thick, and bolt tolerances are tight, they can also carry load. The patch must be thick enough in some cases to accept flush head fasteners. There is a practical limitation on patch thickness and size based on aerodynamic considerations. The patch may have beveled edges to improve aerodynamic conformability.

The bolt pattern will affect the efficiency of a bolted repair. Computer programs have been developed to analyze the bolted repair approach in terms of the number of fasteners and their location.[6]

Another approach to bolted on repair is a flush type patch. In this case, the damage is "cleaned up" and a section is inserted which is now flush with the surrounding undamaged area. A doubler is mandatory with this approach. Fasteners are applied through the patch to the doubler as well as from the undamaged area into the doubler. One of the difficulties associated with the flush type metallic patch is potential difficulty with limited back side access to install large doublers. The large doublers arise from the great amount of fasteners required with such a repair.

Bolted repair techniques use aluminum and titanium as patch materials. Aluminum is lighter than titanium and offers less of a weight penalty. It is also, however, significantly electronegative or anodic to graphite. This leads to possible corrosion problems. Aluminum patches must be physically separated from the underlying composite by scrim cloth or sealant or both. Titanium will not corrode in the presence of graphite. Titanium presents additional difficulty in regards to machinability and formability. Specific titanium alloys are being made available which are easier to form or shape.

Bolted repairs are not without limitations. The drilling operation can be time consuming and improper drilling may introduce additional damage. Also, bolted repairs cannot be used on honeycomb structure.

COMPOSITE SERVICE DAMAGE EXPERIENCE

The earliest application of composites in aircraft was the use of fiberglass reinforced polyester or epoxy structures. These materials were used in radomes, antenna domes, and secondary structures. The types of structures included both simple laminate and laminate faced honeycomb construction. These components, particularly radomes, projected outwards and were subject to a variety of damage including erosion, disbonding, and penetration. In this case, repair was considered in the context of restoring electrical integrity to the component since structural loading was relatively light. With repair of such components, bonded repair approaches were used. Monolithic skin components were repaired with externally bonded patches. Fiberglass faced honeycomb core construction was repaired by core replacement and an external patch. Of prime importance in such repairs was restoration of the electrical transmission characteristics of the components. Thickness variations, for instance, which changed the electrical characteristics could constitute the basis for rejection of the repair.

A similar repair philosophy was used for structural fiberglass composites. Restoration of some measure of mechanical strength was desired, though not critical since these components were not highly loaded. However, since these components were subjected to some mechanical loading, the repair patches were usually thicker and larger than those placed on radomes. Some of the resins used for the repairs were ambient curing systems which also required no additional pressure for compaction. Some components were exposed to elevated temperatures and ambient temperature curing resins were not adequate for the task. In these cases, resistance heating blankets were used to provide the elevated cure temperatures required. Additional compaction was provided by vacuum bagging the repair area.

Other types of reinforcement were subsequently utilized in design of aircraft components. Boron/epoxy (B/epoxy) components were designed in the late 1960's and early 1970's. There was a marked difference between the capabilities of such a fiber and fiberglass. Primary structure was designed taking advantage of the high modulus and strength of the Boron fiber. The repair of such structure had to be approached in a more scientific fashion. Again, there are various classifications of damage ranging from superficial to extensive substructure damage. For cosmetic purposes, the area need only be filled in with a suitable (temperature) two-part system. Through-the-skin damage was pursued by two approaches. One approach used a patch to take the load over the damaged area; the

other used a scarf type repair. In repairing B/epoxy components, field repair limitations on damage size were found. At the field level, a patch approach was developed to effect the repair. Pretreated Ti foil was used in a stacked arrangement with suitable numbers of layers of film adhesive. Prepreg patches were also developed. Autoclaves were used for more involved procedures or larger damage sizes. Where facilities were available, scarf type repairs could be undertaken. Here the area had to be carefully machined, usually with a router. Although Boron composites were used in some military applications, the relatively high cost of the fiber prevented more extensive use. Nevertheless, it is used today in hybrid designs or other applications where high rigidity is required along with high strength.

With the advent of Boron composites, bolted repair concepts were initiated to some degree. This approach has been pursued for metals and continues on today. The major load carrying uses of Boron composites and to a much greater degree of utilization, Graphite (Gr) reinforced composite components, required an alternate approach to a bonded repair concept. Bolted repairs were really given impetus in the latter half of the 1970's particularly as part of the AV8B development program. The thick Gr/Epoxy composite wing skins and composite skin/substructure of the empennage were ideal applications for bolted repair. The patch configuration was used more frequently than a scarf approach since it was easier to implement. Al and Ti were patch materials of choice for repair of Gr/Epoxy components. Ti or stainless steel fasteners are used on bolted repairs since they are compatible with Gr/Epoxy as well as other composite materials.

Gr/Epoxy and other composite materials can be repaired following either a bolted or bonded repair approach. The reasons, as alluded to earlier, for the selection of a particular approach include, facilities available, type of structure being repaired, time to effect the repair, personnel training and experience as well as other factors.

VALIDATION OF REPAIR CONCEPTS

Bolted Metallic Patch Repair

One program demonstrated the use of mechanically fastened titanium patches for repair of Gr/Epoxy wing skins.[7] In this case the repair was required to be performed with only outside access to the aircraft, with a minimum in terms of training, and with the available equipment. Skin thicknesses ranged from .19 in. to .50 in. and damages up to 4 in. in diameter in the vicinity of substructure were considered. Five specimen configurations were fabricated and tested. Static tension tests were conducted to failure on three specimens with skin damage near substructure. Two pressure box specimens were also fabricated. One of the two incorporated a rib with cap damage, and the other contained a spar/rib intersection with accompanying spar/rib cap damage as shown in figure 2.

A 6Al-4V Ti alloy was used as the patch material. The patch edges were beveled for aerodynamic considerations. Ti .25 in. countersunk single shear Jo-bolt fasteners were used at a spacing distance of 4 times the diameter. Fasteners were located as much as possible in line with the direction of load. Polysulfide sealant was applied to the patch, laminate and fasteners.

All three static tension specimens achieved the design limit strain. The pressure boxes were tested to design limit pressure without damage or leakage.

Bonded Patch Repair

A bonded patch type repair for primary structure was developed as part of an in-house U.S. Navy program. Some of the design constraints included one side access and minimization of out-of-moldline protrusion. The repair designs were directed to full-depth honeycomb structure such as horizontal stabilizers and control surfaces with thin monolithic skins up to about 0.20 in. thickness. Damage consisted of through-the-thickness holes up to 4.0 in. in diameter.

The repair, sketched in figure 3, was made by stacking 0.01 in. thick crossplied discs of AS/3501-6 graphite/epoxy prepreg over a 0.015 in. thick crossplied disc of 7743/2054 fiberglass/epoxy prepreg. The two ply discs were cut to size from crossplied sheets of graphite and fiberglass. A quasi-isotropic patch was formed by alternately orienting the discs at +45° and 0°/90° to the principal load direction. The stacked discs were placed on a sheet of EA 951 adhesive and centered over the damage. The patch was cocured in place under vacuum pressure at 350°F for 120 minutes. The discs were sized such that a taper ratio of 25 to 1 was obtained.

Room temperature static strength restoration of 80 percent of the "B" basis strength allowable with extensional stiffness restoration was set as the design goal. A design temperature range of -67°F to 220°F was established with representative moisture levels of 1 to 1.2 percent. Both monolithic and full-depth honeycomb beam specimens were statically tested. The repaired skin on the beam specimen (3 in. damage) was subjected to compression loads above 80 percent "B" allowance and then tested in tension to failure. The monolithic specimens were subjected to tension loads only. Table I summarizes the major parameters and test results.

TABLE I BONDED REPAIR TEST SUMMARY

Laminate Thickness (In.)	Damage Size (In.)	Failure Strain (μ in./in.)	"B" Allow (%)	Failure Mode
0.10	2	7050	85	Net Tension
0.15	2	6125	75	Net Tension
0.15	3	>7550	>90	Skin-Core bond
0.15	4	5500	66	Patch Unbonding

One specimen achieved 75 percent of the "B" allowable which does not quite satisfy the design goal but is sufficiently greater than the current ultimate design strain levels currently being used (4000 to 5000 μ in/in.) to provide satisfactory margins. For the 4 in. damage specimen, an inadequate bond caused premature failure of the specimen and thus the effectiveness of the patch for this size damage could not be fully evaluated. The poor quality bond may have been caused by outgassing of faulty potting compound.

The effects of moisture on the quality and strength of the repair was evaluated.[8] Panels of 24 and 48 ply material were environmentally exposed to absorb 1.2% moisture. Some specimens were dried for 24 hours at 250°F resulting in an average moisture content of 0.7%. The panels were then subjected to a representative cure cycle with a maximum temperature of 350°F, held for two hours both with and without a patch applied. Non destructive inspection and photomicrographs of the parent laminates without the patch indicated no blisters, microcracks, or voids. This was encouraging since in a previous program[9] using 3501-5 material, blistering was observed. Tests performed in interlaminar shear and flexure showed reductions in general agreement with a previous investigation.[10] Non destructive inspection and photomicrographs of the patched laminate indicated the presence of voids within the patch and bondline. Patches with relatively high levels of porosity tended to fail in the vicinity of the bondline and exhibited significantly reduced properties. Interlaminar shear values at 220°F with 1.2 percent moisture were approximately 55 percent of the room temperature dry values; for 0.7 percent moisture, the values were 65 percent of the room temperature dry values.

Static and fatigue tests of full-scale repairs were performed under hot/wet conditions. Two repaired panels, 0.105 in. thick having a 20-ply patch cocured at 350°F over a 2 in. damage were statically tested. One panel was tested dry at room temperature while the second panel was moisturized to 1.2 percent before and after repair. A 17 percent reduction in strength for the hot/wet condition was obtained. For the fatigue test, a repaired panel 0.19 in. thick was exposed before and after repair to a moisture content of 1.2 percent. The specimen successfully sustained two lifetimes of spectrum tension fatigue and two lifetimes of compression fatigue without failure.

SPECIFIC COMPONENT REPAIR

AH-1 Composite Main Rotor Blade

Repair of composite components has been a concern for rotary wing as well as fixed wing aircraft. An effort[11] was undertaken to establish a repair procedure for the AH-1 composite main rotor blade. This particular blade consists of a filament wound S-glass/epoxy spar, a filament wound Kevlar/epoxy basketweave skin and polyimide honeycomb core afterbody. Three types of repair schemes were used for the afterbody structure. One was a skin patch for repair of small punctures and cuts, another was a plug patch for repair of skin and core damage on one side of the blade and a double plug patch for through hole damage. The last method was a V-shaped double patch for repair of the trailing edge. The skin patch developed consists of a precured 3 ply epoxy laminate. The outer ply was 181 glass fabric with the middle ply being a ±45 bias double ply with unidirectional E glass and the inner ply of 120 glass fabric. Damages up to 7 in. in diameter were repaired with these patches. The patches were bonded over the damaged area.

The repair kit designed for these repairs is a pressure/heat blanket. Pressure of 4 PSI is supplied by a hand pump, and a rheostat controls the heat blanket to 160-180°F. Cure cycles consisted of 15-30 minutes combined heat and pressure and in some cases an additional 30 minutes of pressure alone. Other stipulations were that the maximum adhesive temperature would be 180°F, that the adhesive be amenable to easy mixing and also that it be spreadable. The adhesive had to be thixotropic to facilitate application to the lower surface of the blade. The adhesives chosen after a screening study were Hysol's EA9330 and Hexcel's HP341.

Fatigue tests were performed on repaired root end and outboard blade sections. All repairs on the blade section showed no degradation or propagation of the original damage. Whirl tests were conducted as a final part of the repair substantiation. A 50-hour whirl test was performed on repaired blades. No deviation in blade characteristics was detected, and the repairs showed no tendency to separate or become unbonded.

These repairs were demonstrated on appropriate rotor blades under field conditions. Blades were flown after each repair for a short 10 to 15 minute flight to confirm that the blades had been restored to acceptable status. The field demonstrations indicated that personnel could complete field repairs in not more than 3 hours.

Repair of S-3A Spoilers

A small number of the NARMCO 5209/T300 graphite/epoxy spoilers were fabricated as part of a development program for the S-3A aircraft.[12] One issue was the repairability of such components. There were some general guidelines in this program in that the repair should be made using available materials and be cost effective. It was decided that a scarf patch concept would be used, and the patch would be comprised of matching prepreg material. The patch adhesives were to be 250° curing systems and the EA product (modulus and cross sectional area) of the patch was required to be twice as great as the damaged skin. The diameter of the repair patch was limited so that the average adhesive shear stress would not exceed 850 PSI. The configuration chosen for the patch was a layer of 181 cloth prepreg (used for load introduction from the skin to patch) followed by layers of graphite/epoxy prepreg decreasing in size upward as shown in figure 4. The rectangular configuration was used for scratch damage and the circular for all others. Since the substructure of the spoiler was nomex honeycomb, a core filling epoxy was used for filling in the region previously occupied by damaged core. A 250°F curing film adhesive was used to bond the patch to the spoiler surface.

The spoiler was originally separated into different zones. The zonal location of the damage determined the patch orientation and thickness. The edge configuration did not contain honeycomb and the repair procedure was different for this region. The curing process was accomplished through the use of vacuum heating blankets. These repaired spoilers have been flying on S-3A aircraft and have met all performance requirements.

Repair of the Vertical Fin of the L-1011

Repairs have also been conducted on commercial aircraft such as the L-1011. In this case, a vertical fin from an L-1011 was selected as the item for study.[13] A damage 2 in. x 4 in., simulating a lightning strike on the vertical fin, was made by impacting to obtain delamination and by burning through the skin with a welding rod. The area was then charred by heating with an oxygen acetylene flame torch.

The repair concept used for this effort was based on a precured bonded external graphite/epoxy patch. The patch was sized to match the EA product of the skin and stiffener flanges. No 90° plies were included to avoid a large difference in Poisson's ratio between the patch and fin cover. The length of the patch was sized so that the maximum average shear stress did not exceed 500 PSI. This conservative approach was based on the possibility of moisture in the fin laminate and potential voiding in the bondline. The repair was sized to carry 150% of design ultimate load for the fin cover to ensure it would not fail during residual strength testing. As figure 5 shows, precured 4-ply (45, -45, -45, 45) and 3-ply 0_3 patches were bonded in sequence for the repair. The 4-ply patch was fabricated from Narmco T300/5208 and the 3-ply patch from AS4/3502 prepreg material. All patches were ultrasonically inspected and found to be void free. Before the application of the patch it was necessary to clean out the burned through hole. A precured graphite/epoxy disc was then bonded into the hole with EA9330, a two-part room temperature curing adhesive system. A layer of Narmco Metalbond 329 adhesive was then applied to each surface of the five precured layers and the layers laid up maintaining a 1/4-inch step at each layer. The repair was vacuum bagged and cured at 350°F for 2 hours. The cured patch was inspected with an ultrasonic technique and bondline voiding was detected. Unaugmented vacuum cure was given as the primary reason for the presence of bondline voids. The hat stiffeners were repaired by mechanically attaching hat flanges to the repaired skin after the graphite/epoxy patch was inspected. After the completion of the repair, the vertical fin was subsequently loaded statically until failure. There were no apparent effects of the fatigue loading and the article failed statically at 120% of design ultimate load with failure occurring well away from the repair area.

FUTURE REQUIREMENTS

Bonded and bolted repair offer a viable approach to repair of damaged composite components. Properly designed and processed repairs have restored full structural integrity to damaged components. Nevertheless, there are specific areas which must be addressed to successfully implant this methodology as part of the maintenance philosophy. With bonded repairs, the problem of moisture and subsequent bondline voiding is a continuing concern. Drying prior to repair is not a practicable approach to the problem. Alternative methods must be used to obviate this problem. New materials and processes are required to simplify the repair procedure particularly the remote or field situation. Material storage, cure time, amount of pressure and temperature required are limiting factors in a bonded composite repair process which must be considered.

Bolted repairs can be made considerably more efficient. Current development of new titanium alloys offers improved machinability and formability over current alloys. These new alloys also offer an interim field repair method for high temperature resin composites until improved bonded methods can be developed. The use of these new alloys in concert with new repair designs offer the potential for increasing the size of maximum repairable damage.

Along with implementation of new technology is the requirement for adequate training and certification of personnel. Proper instruction and preparation of responsible individuals is necessary for assigned repair tasks. Preparation and dissemination of the appropriate instruction manuals on the subject is also required.

REFERENCES

1. H.Y. LoKein, "Dupont Aramids for Advanced Composites," Industrial Comp. Seminar, S.A.M.P.E. Midwest Chapter, March 9, 1982.

2. J.M. Whitney, "Composites in the Aircraft Industry," ASTM Standardization News, pp 16-19, December (1983).

3. C.E. Beck and S. Myhre, "Repair Concepts for Advanced Composite Structures," Journal of Aircraft, Vol. 16, No. 10, October 1979, p 720.

4. R.L. Coggeshall, "737 Graphite Composite Flight Spoiler Flight Service Evaluation," NASA Contractor Report No. 165826 of February 1982.

5. J.C. Watson, "Bolted Field Repair of Composite Structures," NADC Report No. 77019-30 of 1 March 1979.

6. R.E. Bohlmann, G.D. Renieri and M. Libeskind, "Bolted Field Repair of Gr/Epoxy Wing Skin Laminates," ASTM Symposium — Joining of Composites, Minn., Mn., 15-16 April 1980.

7. T. Kanno, "Review of The Navy Composite Repair Program," 26th National S.A.M.P.E. Symposium, 28-30 April 1981.

8. E.L. Rosenzweig, "Environmental and Fatigue Evaluation of Field Repair Concepts," NASC Workshop on Shipboard Repair of Composite Structural Materials, 9-11 September 1980.

9. J.D. Labor and S.H. Myhre, "Large Area Composite Structure Repair," AFFDL-TR-79-3040.

10. G. Hedrick and J. Whiteside, "Effects of Environment on Advanced Composite Structures," AIAA Reprint 77-463, March 1977.

11. A.S. Falcone, "Field Repairs of the AH-1 Composite Main Rotor Blade," 11th National S.A.M.P.E. Conference, November 13-15, 1979.

12. C. Foreman, S.A. McGovern and R. Knight, "S-3A Gr/Epoxy Spoiler Fabrication of Ten Shipsets and Damage Repair Study," NADC Report No. 76234-30 of May 1976.

13. R. Stone, "Development, Demonstration and Verification of Repair Techniques and Processes for Gr/Epoxy Structures for Commercial Transport Applications," NASA Contractor Report No. 159056 of January 1983.

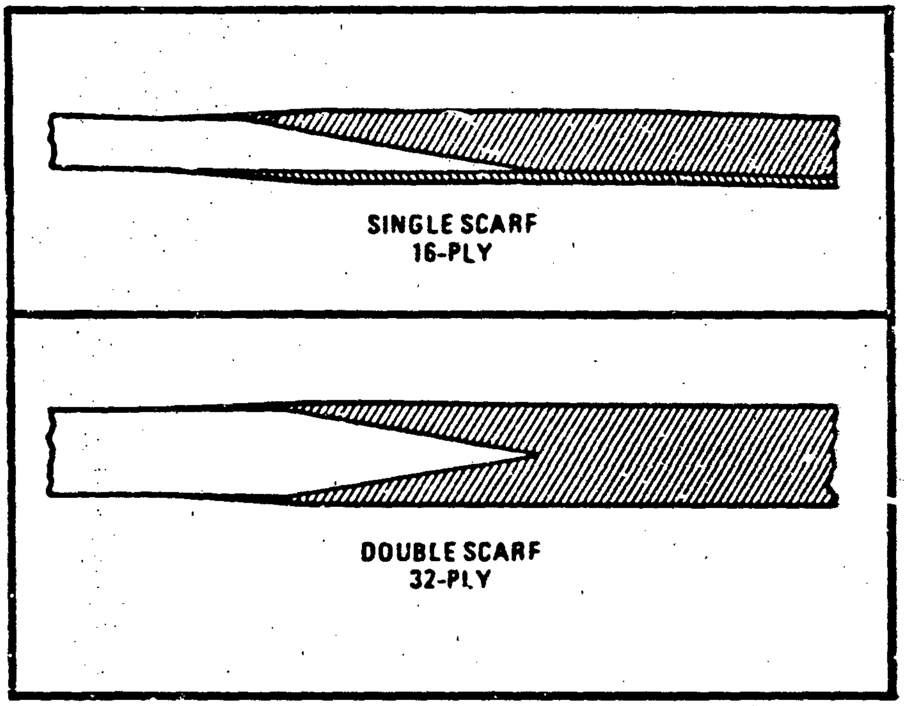

Figure 1. Single and Double Scarf Repairs

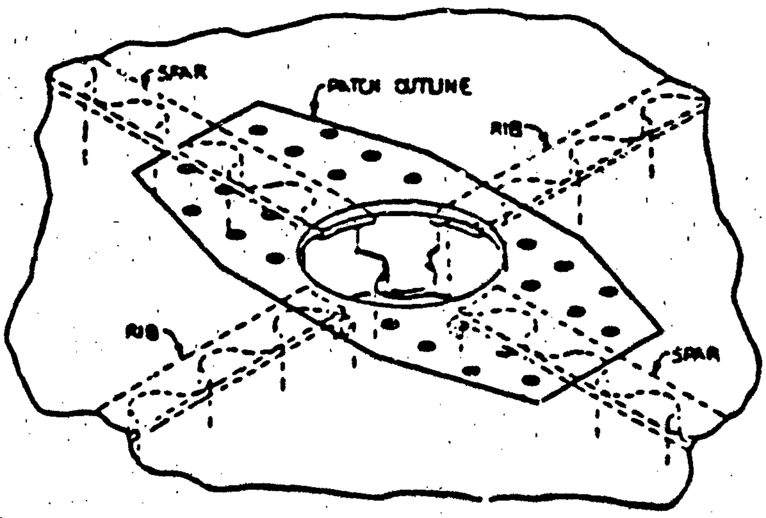

Figure 2. Spar Rib Cap Damage

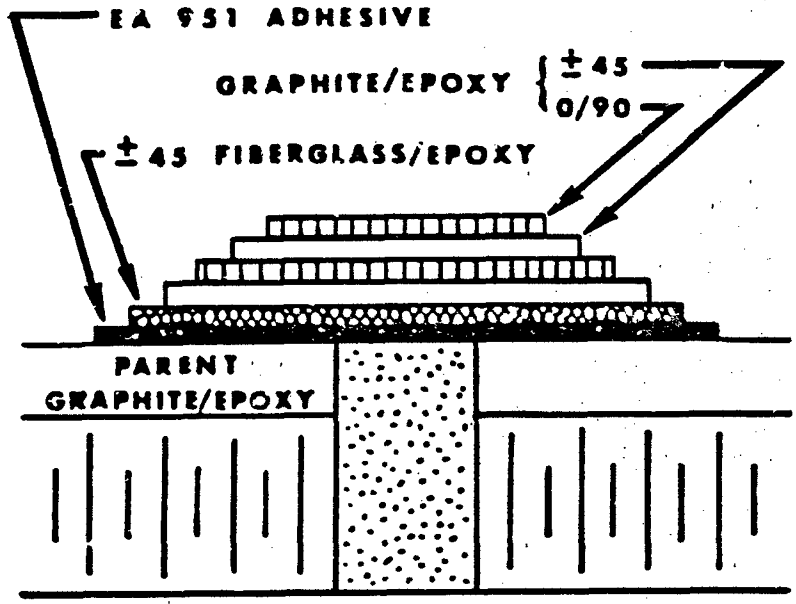

Figure 3. Honeycomb Repair Specimen

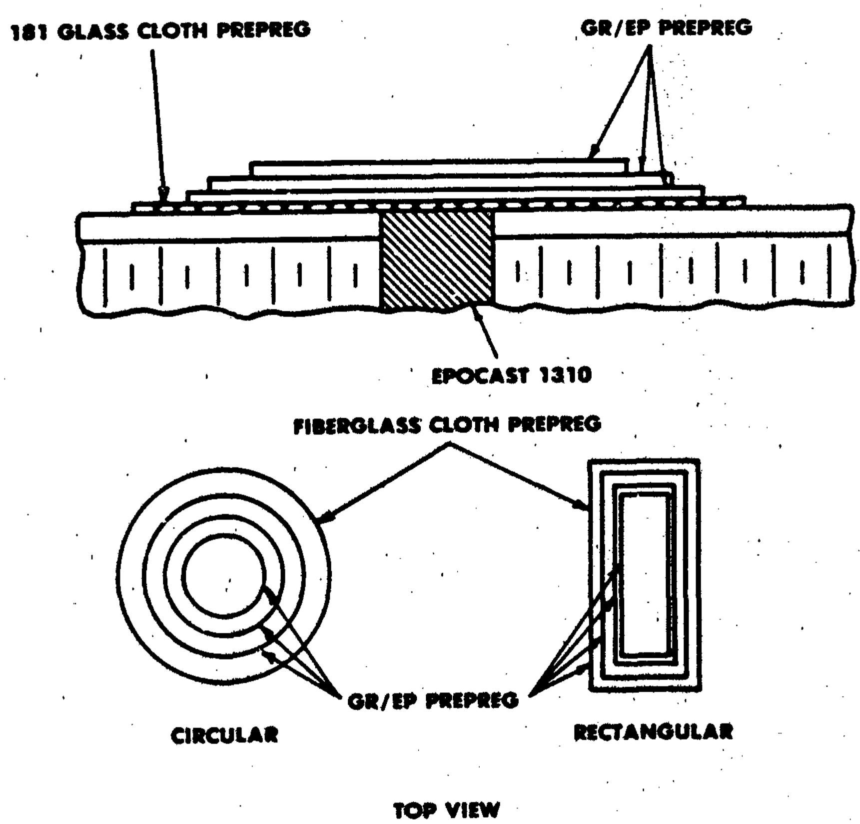

Figure 4. S-3A Spoiler Repair Patch Configuration

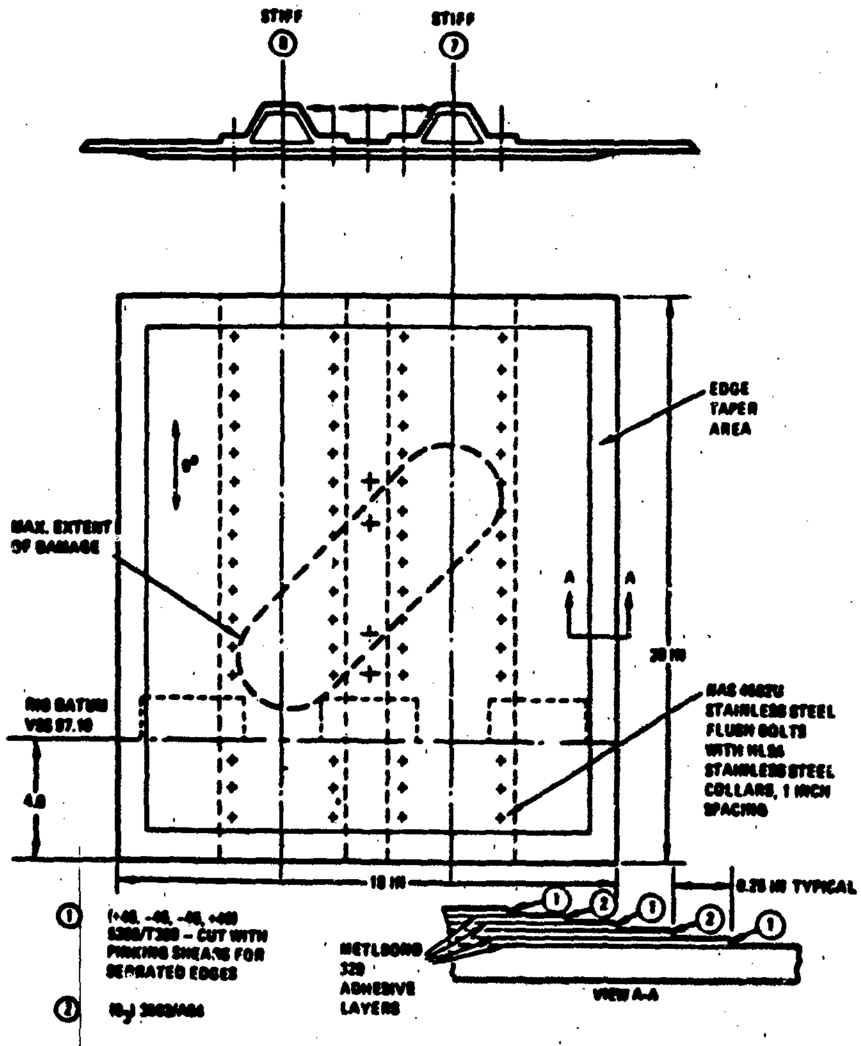

Figure 5. Repair Configuration for Damaged L-1011 Fin

COMPOSITE REPAIR OF AIRCRAFT STRUCTURES

Josef W. Vilsmeier

MESSERSCHMITT-BÖLKOW-BLOHM GMBH
Postfach 80 11 60
8000 München 80

SUMMARY

The repair of composite and metallic aircraft structures with composite materials has become a matter of "world wide interest and activity".

This is due mainly to the fact that in modern A/C structures the amount of composite parts are steadily increasing.
This paper presents the activities of MBB in the field of composite repair of flat laminates, honeycomb sandwich structures and integral stiffened panels. In the second part of the paper on composite repair problems, which have not yet been fully investigated, Aircraft Battle Damage Repair (ABDR), bonding on wet laminates, environmental effects, NDT problems of repaired structures and repair of joints are discussed.

INTRODUCTION

The first paper on Composite Structure Repair, presented before this AGARD panel by Mr. Kelly from AFWAL/FIBC, is an excellent overview about two repair configurations that are generic in nature (external patch and near flush repair). He has shown that these methods of repair have been verified in the U.S. for a wide varity of light to moderately bonded, stiffened and honeycomb sandwich structures.
This paper has two major objectives. In the first part the activities on Composite Repair of A/C-structures at MBB will be presented, while in the second part some problems on Composite Repair, which have to be solved to give a more complete understanding, will be discussed.

ACTIVITIES ON COMPOSITE REPAIR OF A/C-STRUCTURES AT MBB

OVERVIEW

Currently three typical types of composite repair are investigated at MBB:
- repair of flat laminates
- repair of honeycomb sandwich structures
- repair of integrally stiffened panels.

The parent material which has been used for these investigations is Fibredux 914C/T300 from Ciba Geigy. The other materials used will be described together with the repair methods.

The design criteria which have been applied to the repair components are as follows:

- 100% ultimate design strength recovery without any appreciable change in stiffness
- Serviceability over a temperature range from -55°C to 110°C
- Minimum weight penalty
- Minimum aerodynamic contour changes
- Amenability to rework in the factory, depot and in the field

In most cases, the approaches MBB has used for the repairs are very similar to the solutions which are already known from literature, because we want to verify already approved repair concepts for our materials and configurations.

The ultimate strain allowables are approximately the same as those used in the US (~ 4000 μin/in). This allows for stress concentrations due to bolt holes and invisible, low energy impact damage. This means, that the repair configuration should show, for reasons of safety, somewhat higher ultimate strain figures than the design allowables of the parent laminate. If a bolted structures has to be repaired in a region away from the bolts, the failure should occur in the bolted laminate and not in the repair patch, or repair bond line.

REPAIR OF FLAT LAMINATES

The standard repair on flat laminates was chosen to be similar to a depot level repair. This means that the parent materials and the repair material should be basically the same and a structural adhesive with a 175°C cure should be used. There should be the facilities to apply vacuum pressure equal or better than 0,5 bar and temperatures up to 190°C. The parameters for the repair of the flat laminates are as follows:
- influence of different types of laminates
- influence of the diameter of the repair
- influence of the laminate thickness
- influence of supplementary external layers.

The specimen geometry and the types of lay-up can be seen in Fig. 1 and in Fig. 2.

The repair was done for all specimens using a taper ratio of 20:1. After the taper was milled a layer of adhesive Redux BSL 319 was applied on the tapered surface and the repair material, which is identical with the parent material, was built up in the scarf and co-cured with 0,5 bar vacuum pressure at 175°C and postcured for 4h at 190°C.

The void content of the repair laminate, measured on a manufacture control specimen, was found to be about 1,5-2,5%. The specimen will be tested in tension. The test conditions will be RT, "as received" and 120°C, 1.2% moisture content.
The moisture content will be monitored by a traveller specimen which is representative of the parent laminate and humidifying will be done at 70°C and about 75-85% R.H.
A test table is shown in Fig. 2.

REPAIR OF HONEYCOMB SANDWICH STRUCTURES

The damage size of the honeycomb sandwich is a hole with a diameter of 50 mm. This is thought to be representative for a hole damage resulting from ballistic or foreign object penetration. The crushed honeycomb was cut out to the full depth and replaced by a Rohacell foam (Type 110).
The types of repairs are: a depot repair which is flush; two types of field repair, one being an external patch with Titanium foils, the second one has an external patch, but the patch material is graphite epoxy. The main characteristics of the selected repairs can be seen in figure 3. The parent laminate consists of 12 plies (2/2/8) of Fibredux 914C/T300 (2-0° plies, 2-90° plies, 8- ±45° plies). Only one side is repaired, therefore the laminate on the back side has 16 plies (6/2/8) to prevent failure in this laminate. The repair patch is always a orthotropic lay-up of graphite epoxy.

The test matrix is seen in figure 4. All specimens are tested at room temperature in the as received condition. The type of loading is tension or compression in four point bending. One half of the specimens are subjected to a static test, the other half of the specimens will be subjected to two lifetimes of spectrum fatigue loading and the residual strength will be established after fatigue.
The specimen geometry of the repair configurations can be seen in figures 5-8.

REPAIR OF INTEGRALLY STIFFENED PANELS

The repair of integrally stiffened panels presents a much more complex situation than the preceding repair problems. To avoid further complication it was assumed, that the accessability to the repair is from both sides. Again, out of several possible solutions the most promising for field and depot-level repairs have been chosen. Figure 9 presents the typical features of the "J"-stiffener repair. The damaged zone is represented by a cut-out of 50 mm diameter which contains both the "J"-stiffener and the skin laminate. A block of equal shaped Rohacell foam is bonded into this cut-out and the foam is shaped to the contours of the stiffener. The depot repair is made with the same material as the parent laminate. The field repair is basically a wet lay-up with carbon fibre fabric and EA 956 resin. The principle configuration of the "J"-stiffened panel is shown in figure 10. The cross-sections and side-views of the repairs can be identified in figure 11 for the depot repair and in figures 12 for the field repair. The test conditions for the unrepaired and repaired panels will be: static "as received"; two life times spectrum fatigue loading and a residual static strength test at the end of fatigue loading. These tests will be conducted at RT. There will be a combined loading for the test of these panels, torsion and internal pressure which is relevant for this type of structural element. A schematic diagram of the test box is shown in figure 13. If the test at room temperature shows good results, those of the repair configurations, which are supposed to be applicable for permanent repairs, will then be tested in a hot/wet condition.
The next step would be the repair of a stiffened panel with accessability from one side only. Until now, we have not investigated this problem in detail.

OPEN PROBLEMS

AIRCRAFT BATLLE DAMAGE REPAIR (ABDR)

ABDR presents a big challenge for engineers. A lot of different requirements for ABDR or field level repairs are known.
The time for this type of repair is ranging between 4 and 36 hours.After the ABDR, the aircraft should be able to fly either on more mission or to the maintenance depot.
There are requirements that the full structural strength should be recovered. On the other hand, a structural limitation might be allowed if the repair time to recover the full strength is too long.
In the FRG there will be a small programme for composite repair of A/C-structures with ABDR conditions. It is the aim of this study to prove the applicability of composite repair to metallic structures. Metal and composite repair of metallic structures will be compared with regard to the rapidity and structural effectivity.
In a second programme the possibilities of composite or metallic repair of composite structures under ABDR conditions will be investigated.
As the time for the repair is the most important parameter for the ABDR condition, there is a need for a fast-curing resin, both for bonding purposes, and for wet lay-up of composite fibre fabrics. In the ABDR the repair patch and the parent material are unlikely to be the same. There is a demand for fast-curing epoxies in other industries (.e.g. automative). It would be worth investigating these systems for ABDR applications.

BONDING ON WET LAMINATES

During service, composite structures with an organic matrix are always subjected to moisture. The amount of moisture in the parent laminate is a function of the environmental history and the service life. The entrapped moisture in the parent laminate can impede the repair job.
There is a possibility of blistering in the parent material causing bad bondlines between parent and repair laminates or even of debonding in honeycomb structures. Before starting the repair job the first problem is, that there is no proven NDT method to evaluate the amound of moisture in the parent laminate. Secondly, the re-drying of the parent laminate is very time consuming and the temperature of drying has to be chosen carefully to avoid thermal spiking of the parent laminate. In no case can thermal spiking effects be allowed in permanent repairs, because of the possibility of severe degradation of the parent laminate. There is a figure for an allowable residual moisture content of 0.5 percent before the repair, but this figure is empirically found for one system and perhaps detailed investigations have to be made for other prepreg systems.
A further concern is that the expelled water can cause a bad bondline. This means that porosity can significantly deteriorate the strength of the bonding and therefore the whole repair is of poor quality. Not only the bondline can be affected by the moisture, the repair laminate may also suffer from this effect. This problem can be regarded as less important because the repair laminates will be cured in most cases with vacuum only and, therefore, the quality of these laminates will be not so good as that of the parent laminate.

ENVIRONMENTAL EFFECTS

Environmental degradation due to moisture has to be considered in the case of permanent repairs. A reduction of about 20% in strength allowables was found for repair laminates and perhaps bond strength. It has to be assured, that the environmental degradation of the repair is not greater than that of the parent laminates.
The environmental problem will be of great concern in using 125°C curing prepreg systems and adhesives or room temperature curing systems. For permanent repairs no reduction in serciveability with regard to the maximum design temperature and the design life of the aircraft can be allowed. Therefore the repair methods have to be evaluated with regard to the design conditions.

NDT TECHNIQUES FOR REPAIRED STRUCTURES

As shown above the quality of bondlines and of the repair laminates will be inferior compared to the materials in the unrepaired structures. Therefore, it is very important to have a good control of the quality of the repair patches and the repair bondlines. Unfortunately, the possibilities for NDT inspections on repaired structures are even more limited than those after manufacture of structural parts.

In most cases NDT of repaired parts will have to be done on the aircraft. Therefore US-inspection methods like through transmission and reflector plate technique are not applicable. The design of structural repairs (tapers, patches, etc.) is leading to configurations whereparent laminate, bondline and repair laminate have to be tested together. This means that the detection of critical defects in the bondlines of repair patches is very complicated and the probability for proper detection is low.
Most of the approved NDT-techniques which can be used in the field are done by hand. This means, that in practice the resolution is poorer than of automised NDT methods. Further detrimental conditions for inspection of repairs can be accessabiliy and limitations due to security problems (X-ray).
It is not sufficient to improve repair techniques alone, NDT-methods also have to be developed and improved.

REPAIR OF JOINTS

The repair of joints can be divided into two major problems:
- Bonded joints of metal parts with composite structures
- Bolted joints.

At present, there is little literature available about these problems. The repair of bonded joints of metallic and composite parts is mainly a problem of the surface treatment of the metallic surface prior to bonding. The surface treatment of metal parts is complicated enough without the repair situation.

It has not yet been shown whether the approved surface treatments for bonding of metallic structures are applicable in the repair situation. Some work has been done with respect to "field" surface treatment of clean aluminium surfaces. For titanium a similar process has not been demonstrated.
The specific problem of bolted joint repair is that of a worn fastener hole. Simply refilling worn holes with resin and redrilling is not adequate for a permanent repair. As worn holes are always an indication that the design of the part is inadequate. The repair problem becomes more difficult as a greater strength is required for the bolted joint.
A similar problem occurs for impact damage close to fasteners. It is not desirable that the scarf, which is necessary to remove the impacted material, extends into the bolted area. For both situations a development and validation of acceptable fastener hole repair is needed.

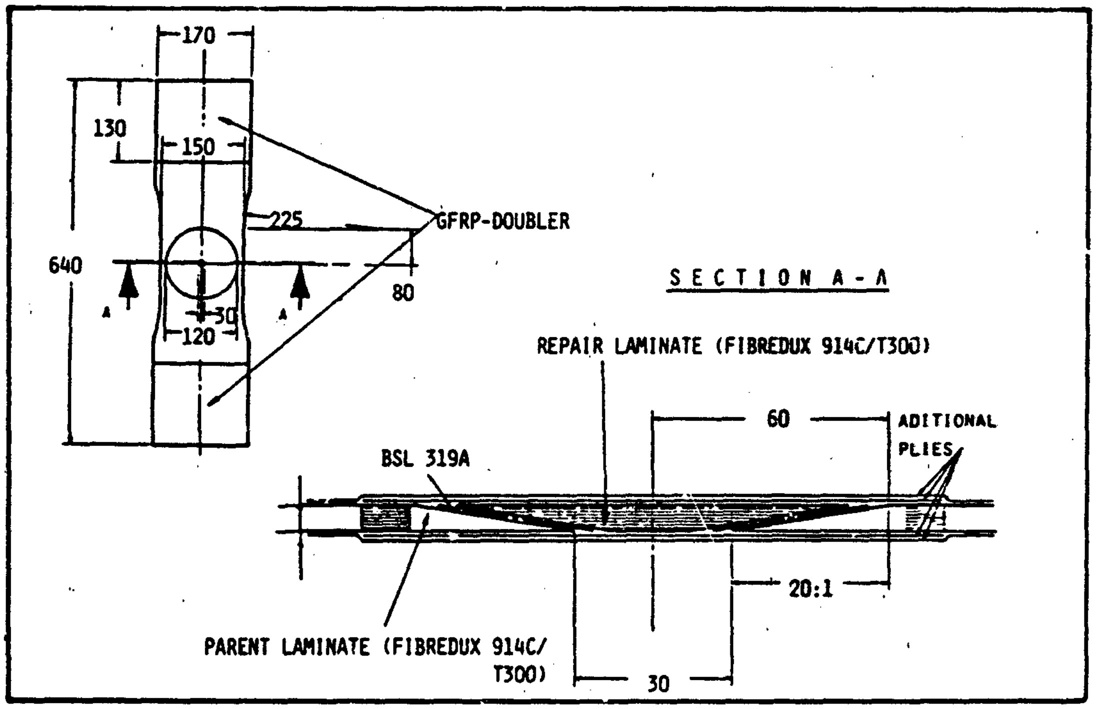

Fig. 1: Specimen configuration - Repair of flat laminates

LAMINATE (%0°/%90°/%±45°)	SPECIMEN THICKNESS	SPECIMEN WITOUT REPAIR RT, DRY	120°C, WET	REPAIRED SPECIMEN RT, DRY	120°C, WET	DIAMETER OF REPAIR
(44.4/11.2/44.4)	2.25	1	4	1	4	100
(44.4/11.2/44.4)	2.25	1	4	1	4	120
(44.4/11.2/44.4)	2.25	-	-	1	4	120 *
(25/25/50)	2.0	1	4	1	4	110
(8.3/8.3/83.4)	3.0	1	4	1	4	150
(25/25/50)	3.0	1	4	1	4	150

* TWO ADDITIONAL ±45° LAYERS ON THE UPPER AND LOWER SIDE OF THE SPECIMEN

Fig. 2: Test matrix - Repair of flat laminates

STRUCTURE	LEVEL	TYPE	FILLER MATERIAL	PATCH MATERIAL	PATCH-TO-SKIN ADHESIVE	PATCH CURING TEMPERATURE	PATCH CURING PRESSURE
HONEYCOMB	DEPOT	INTERNAL	ROHACELL	FIBREDUX 914C/T300	BSL 319A	175°C	VACUUM
HONEYCOMB	FIELD	EXTERNAL	ROHACELL	FI FOIL	EA 956 / EA 934	RT	VACUUM
HONEYCOMB	DEPOT/ FIELD	EXTERNAL	ROHACELL	FIBREDUX 914C/T300	BSL 319A	175°C	VACUUM

Fig. 3: Features of the honeycomb sandwich repair

CONFIGURATION	STATIC RT, DRY	SPECTRUM FATIGUE RT, DRY*	TYPE OF LOAD
HONEYCOMB SANDWICH			
(1) UNDAMAGED	1	1	TENSION IN
(2) DEPOT REPAIR	1	1	FOUR POINT
(3) FIELD REPAIR (Ti-FOIL)	1	1	BENDING
(4) EXTERNAL REPAIR	1	1	
"J"-STIFFENED PANEL			COMBINED
(1) UNDAMAGED	1	1	LOADING:
(2) DEPOT REPAIR	1	1	TORSION AND
(3) FIELD REPAIR	1	1	PRESSURE

* RESIDUALS AFTER TWO LIFTIMES OF FATIGUE TESTING

Fig. 4: Test matrix - Repair of honeycomb sandwich and "J"-stiffened panels

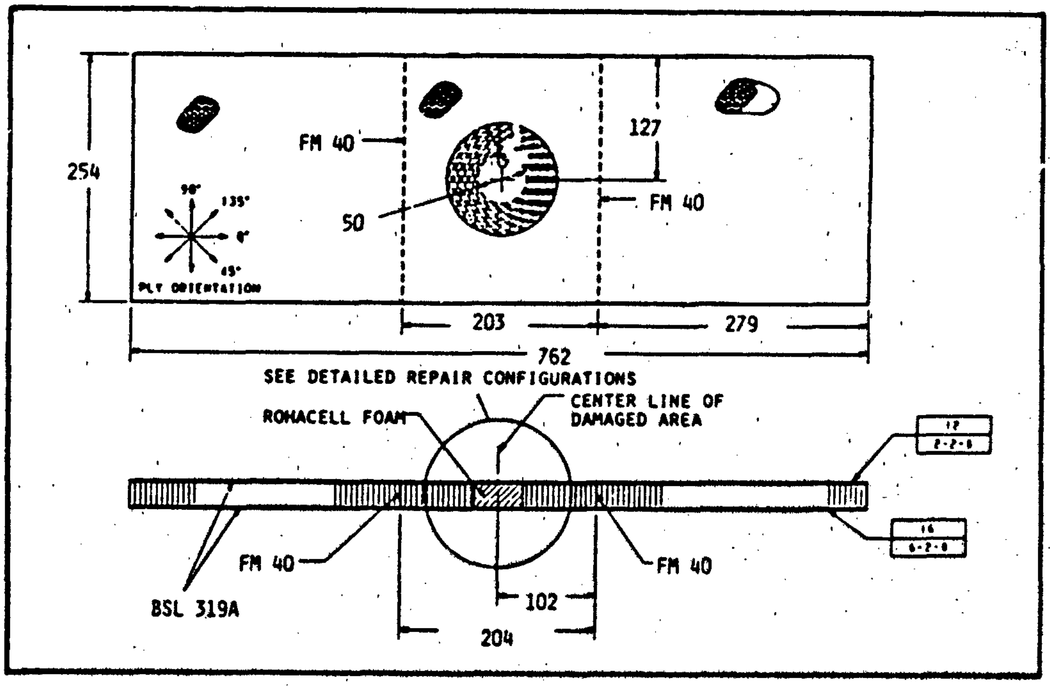

Fig. 5: Configuration of the honeycomb sandwich specimens

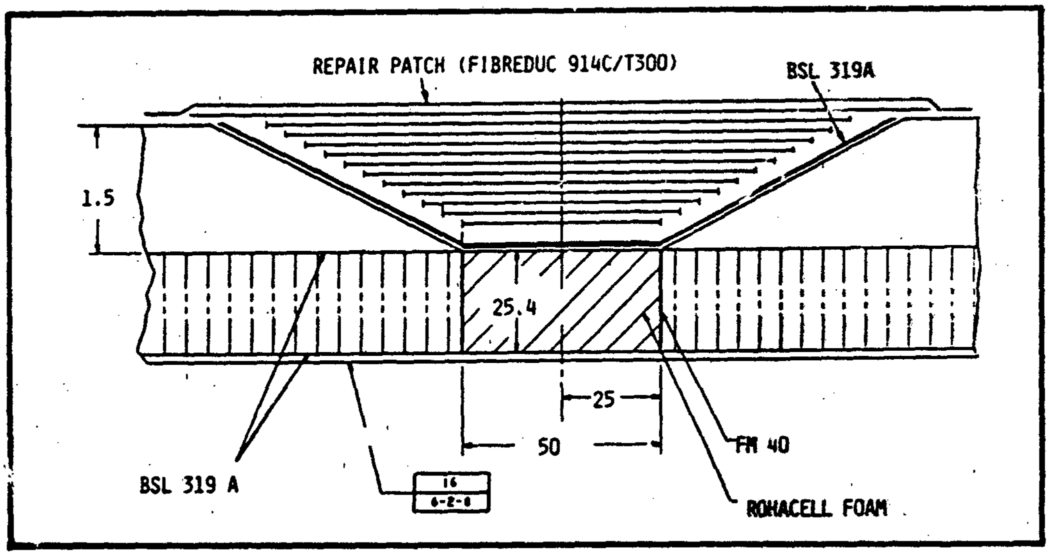

Fig. 6: Depot repair of honeycomb sandwich specimen

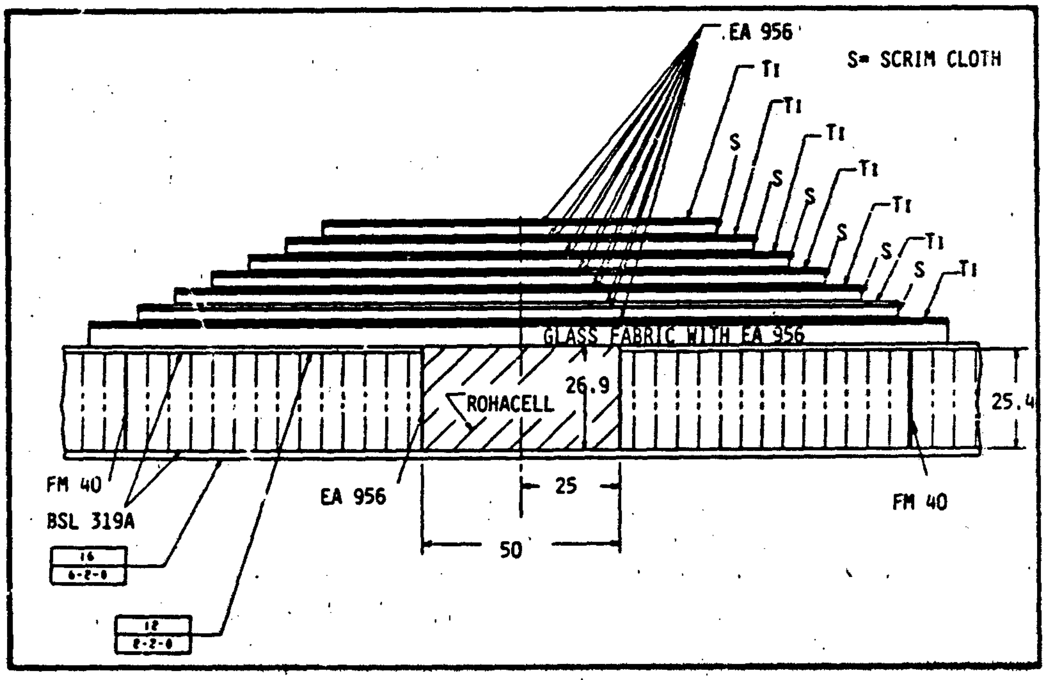

Fig. 7: Field repair of honeycomb sandwich specimen

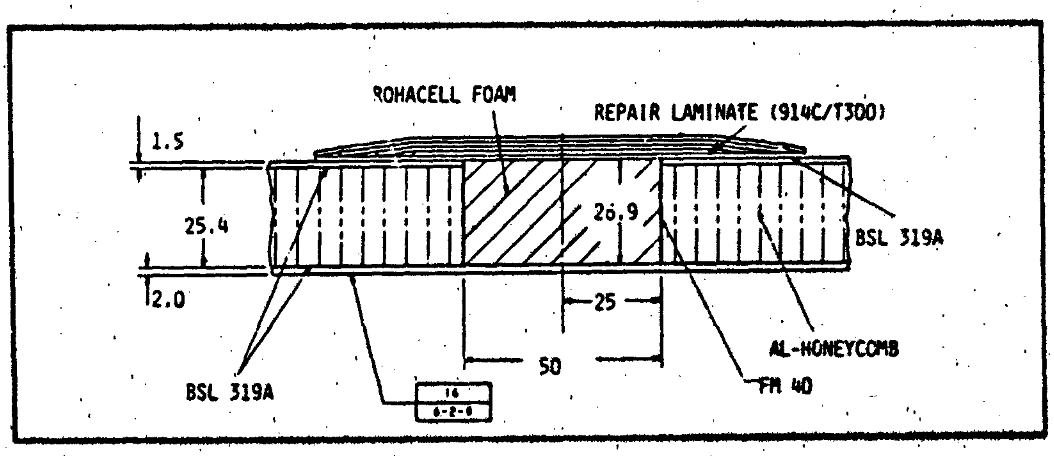

Fig. 8: Depot (field) repair of honeycomb sandwich specimen

STRUCTURE	LEVEL	TYPE	FILLER MATERIAL	PATCH MATERIAL	PATCH-TO-SKIN ADHESIVE	PATCH CURING TEMP.	PATCH CURING PRESSURE
"J"-STIFFENER	DEPOT	INTERNAL	ROHACELL	FIBREDUX 914C/T300	BSL 319A	175°C	VACUUM
"J"-STIFFENER	FIELD	EXTERNAL	ROHACELL	GRAPHITE FABRIC EA 956	EA 956	RT	VACUUM

Fig. 9: Features of the "J"-stiffener repair.

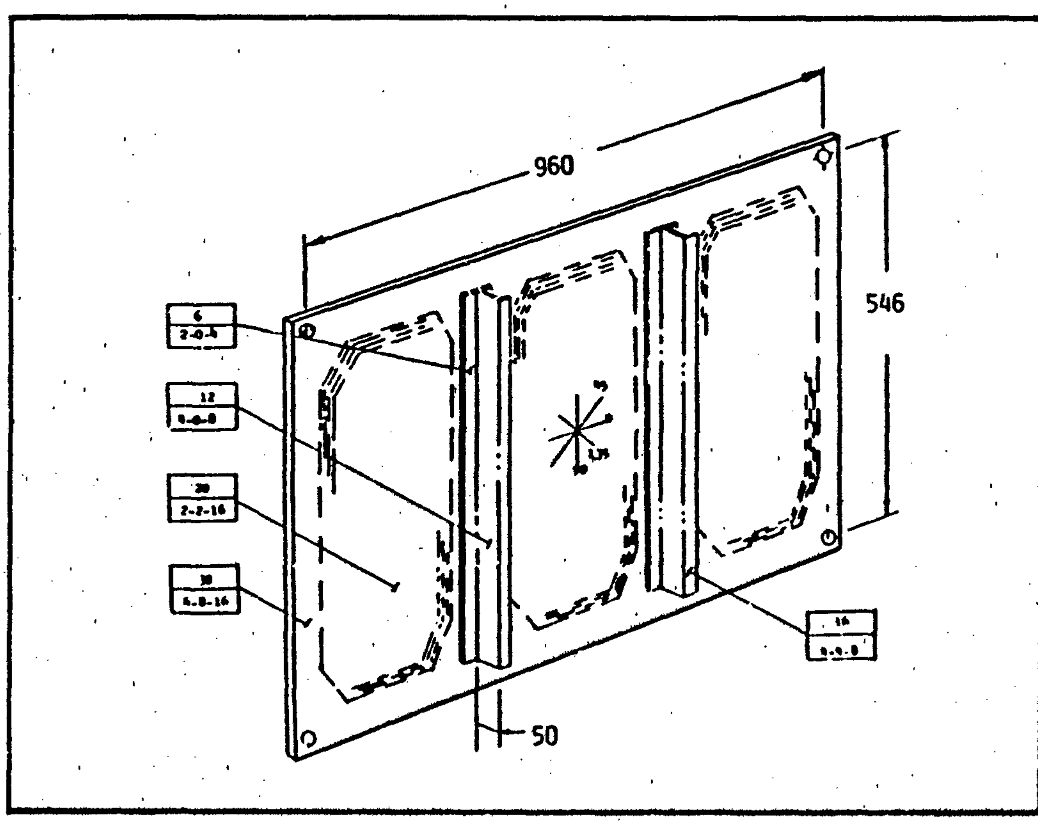

Fig. 10: Configuration of the "J"-stiffened panel.

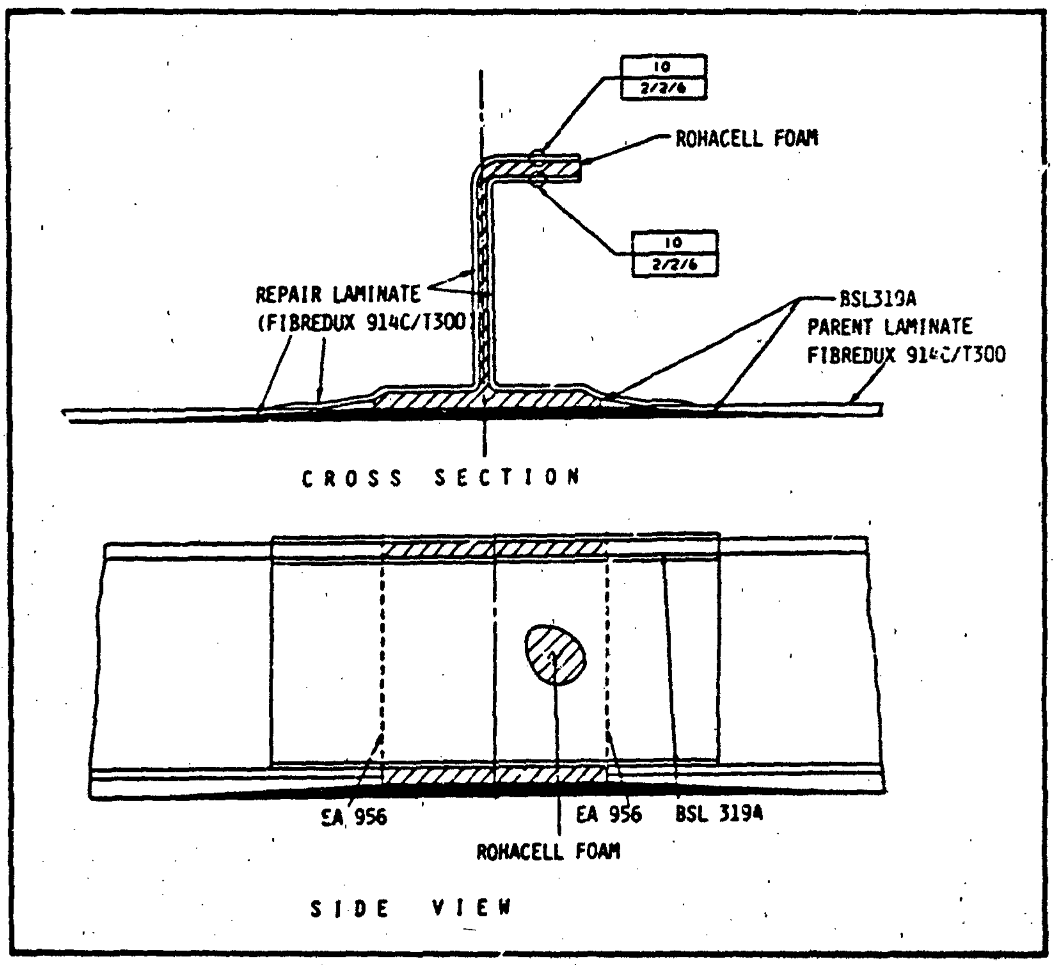

Fig. 11: Depot repair of the "J"-stiffened panel.

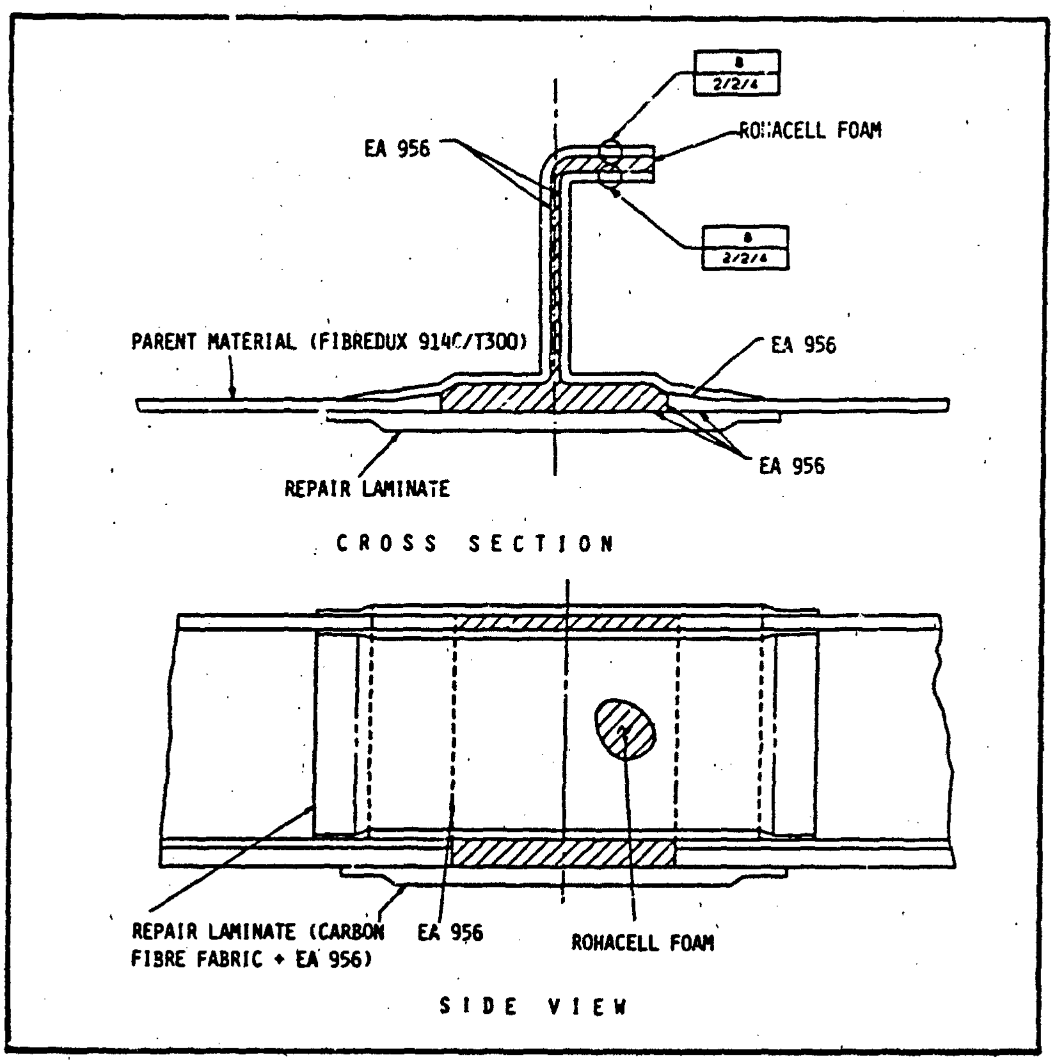

Fig. 12: Field repair of the "J"-stiffened panel.

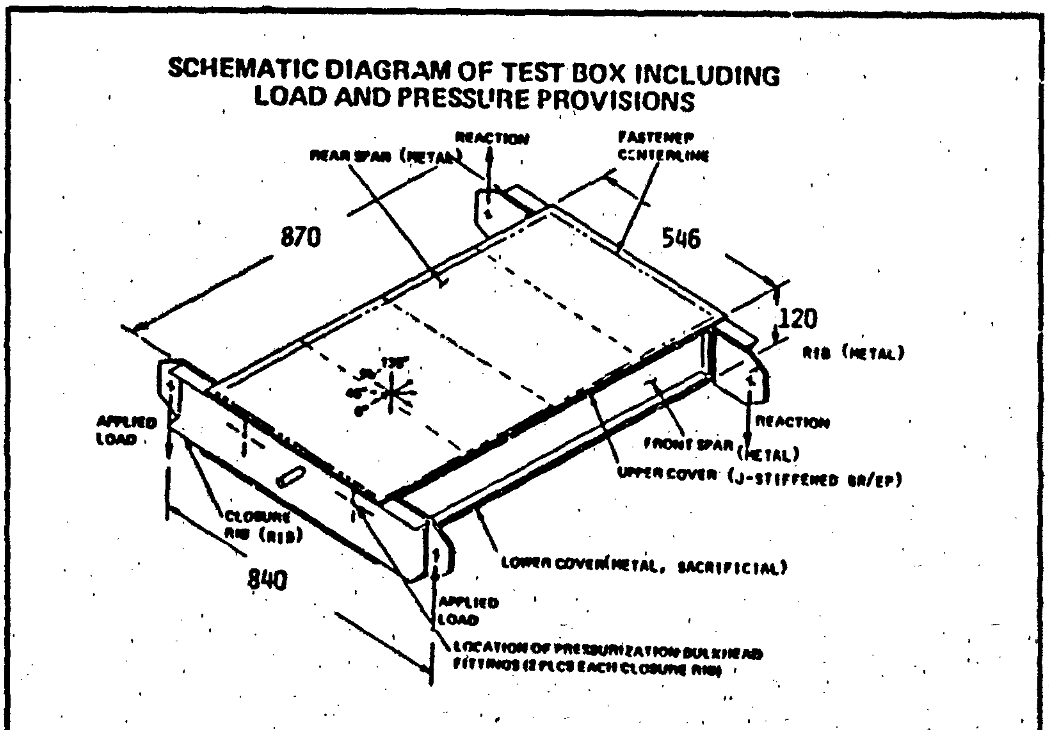

Fig. 13: Test box for the "J"-stiffened panels.

REPORT DOCUMENTATION PAGE			
1. Recipient's Reference	**2. Originator's Reference** AGARD-R-716 Addendum	**3. Further Reference** ISBN 92-835-1476-9	**4. Security Classification of Document** UNCLASSIFIED
5. Originator	Advisory Group for Aerospace Research and Development North Atlantic Treaty Organization 7 rue Ancelle, 92200 Neuilly sur Seine, France		
6. Title	COMPOSITE STRUCTURE REPAIR		
7. Presented at	the 58th Meeting of the Structures and Materials Panel, in Sienna, Italy, on 2—6 April 1984		
8. Author(s)/Editor(s)	Various		**9. Date** August 1984
10. Author's/Editor's Address	Various		**11. Pages** 24
12. Distribution Statement	This document is distributed in accordance with AGARD policies and regulations, which are outlined on the Outside Back Covers of all AGARD publications.		
13. Keywords/Descriptors	Composite materials Composite structures	Maintenance	

14. Abstract

The Structures and Materials Panel of AGARD has been concerned for some time with the characteristics of composite structures and has initiated an activity on their repair. A pilot paper AGARD R-716 entitled "Composite Structure Repair" has already been published; this document includes two further papers recently presented on the topic.

The authors cover current activities in their particular fields and indicate how similar problems are approached by different organisations.

AGARD Report 716 Addendum	AGARD-R-716
Advisory Group for Aerospace Research and Development, NATO COMPOSITE STRUCTURE REPAIR Published August 1984 24 Pages The Structures and Materials Panel of AGARD has been concerned for some time with the characterisitics of composite structures and has initiated an activity on their repair. A pilot paper AGARD R-716 entitled "Composite Structure Repair" has already been published; this document includes two further papers recently presented on the topic. The authors cover current activities in their particular P.T.O	Composite materials Composite structures Maintenance

AGARD Report 716 Addendum	AGARD-R-716
Advisory Group for Aerospace Research and Development, NATO COMPOSITE STRUCTURE REPAIR Published August 1984 24 Pages The Structures and Materials Panel of AGARD has been concerned for some time with the characterisitics of composite structures and has initiated an activity on their repair. A pilot paper AGARD R-716 entitled "Composite Structure Repair" has already been published; this document includes two further papers recently presented on the topic. The authors cover current activities in their particular P.T.O	Composite materials Composite structures Maintenance

AGARD Report 716 Addendum	AGARD-R-716
Advisory Group for Aerospace Research and Development, NATO COMPOSITE STRUCTURE REPAIR Published August 1984 24 Pages The Structures and Materials Panel of AGARD has been concerned for some time with the characterisitics of composite structures and has initiated an activity on their repair. A pilot paper AGARD R-716 entitled "Composite Structure Repair" has already been published; this document includes two further papers recently presented on the topic. The authors cover current activities in their particular P.T.O	Composite materials Composite structures Maintenance

AGARD Report 716 Addendum	AGARD-R-716
Advisory Group for Aerospace Research and Development, NATO COMPOSITE STRUCTURE REPAIR Published August 1984 24 Pages The Structures and Materials Panel of AGARD has been concerned for some time with the characterisitics of composite structures and has initiated an activity on their repair. A pilot paper AGARD R-716 entitled "Composite Structure Repair" has already been published; this document includes two further papers recently presented on the topic. The authors cover current activities in their particular P.T.O	Composite materials Composite structures Maintenance

AGARD

NATO OTAN

7 RUE ANCELLE · 92200 NEUILLY-SUR-SEINE

FRANCE

Telephone 745.08.10 · Telex 610176

DISTRIBUTION OF UNCLASSIFIED AGARD PUBLICATIONS

AGARD does NOT hold stocks of AGARD publications at the above address for general distribution. Initial distribution of AGARD publications is made to AGARD Member Nations through the following National Distribution Centres. Further copies are sometimes available from these Centres, but if not may be purchased in Microfiche or Photocopy form from the Purchase Agencies listed below.

NATIONAL DISTRIBUTION CENTRES

BELGIUM
Coordonnateur AGARD — VSL
Etat-Major de la Force Aérienne
Quartier Reine Elisabeth
Rue d'Evere, 1140 Bruxelles

CANADA
Defence Scientific Information Services
Dept of National Defence
Ottawa, Ontario K1A 0K2

DENMARK
Danish Defence Research Board
Ved Idraetsparken 4
2100 Copenhagen Ø

FRANCE
O.N.E.R.A. (Direction)
29 Avenue de la Division Leclerc
92320 Châtillon

GERMANY
Fachinformationszentrum Energie,
Physik, Mathematik GmbH
Kernforschungszentrum
D-7514 Eggenstein-Leopoldshafen

GREECE
Hellenic Air Force General Staff
Research and Development Directorate
Holargos, Athens

ICELAND
Director of Aviation
c/o Flugrad
Reykjavik

ITALY
Aeronautica Militare
Ufficio del Delegato Nazionale all'AGARD
3 Piazzale Adenauer
00144 Roma/EUR

LUXEMBOURG
See Belgium

NETHERLANDS
Netherlands Delegation to AGARD
National Aerospace Laboratory, NLR
P.O. Box 126
2600 AC Delft

NORWAY
Norwegian Defence Research Establishment
Attn: Biblioteket
P.O. Box 25
N-2007 Kjeller

PORTUGAL
Portuguese National Coordinator to AGARD
Gabinete de Estudos e Programas
CLAFA
Base de Alfragide
Alfragide
2700 Amadora

TURKEY
Department of Research and Development (ARGE)
Ministry of National Defence, Ankara

UNITED KINGDOM
Defence Research Information Centre
Station Square House
St Mary Cray
Orpington, Kent BR5 3RE

UNITED STATES
National Aeronautics and Space Administration (NASA)
Langley Field, Virginia 23365
Attn: Report Distribution and Storage Unit

THE UNITED STATES NATIONAL DISTRIBUTION CENTRE (NASA) DOES NOT HOLD STOCKS OF AGARD PUBLICATIONS, AND APPLICATIONS FOR COPIES SHOULD BE MADE DIRECT TO THE NATIONAL TECHNICAL INFORMATION SERVICE (NTIS) AT THE ADDRESS BELOW.

PURCHASE AGENCIES

Microfiche or Photocopy
National Technical
Information Service (NTIS)
5285 Port Royal Road
Springfield
Virginia 22161, USA

Microfiche
ESA/Information Retrieval Service
European Space Agency
10, rue Mario Nikis
75015 Paris, France

Microfiche or Photocopy
British Library Lending
Division
Boston Spa, Wetherby
West Yorkshire LS23 7BQ
England

Requests for microfiche or photocopies of AGARD documents should include the AGARD serial number, title, author or editor, and publication date. Requests to NTIS should include the NASA accession report number. Full bibliographical references and abstracts of AGARD publications are given in the following journals:

Scientific and Technical Aerospace Reports (STAR)
published by NASA Scientific and Technical
Information Branch
NASA Headquarters (NIT-40)
Washington D.C. 20546, USA

Government Reports Announcements (GRA)
published by the National Technical
Information Services, Springfield
Virginia 22161, USA

Printed by Specialised Printing Services Limited
40 Chigwell Lane, Loughton, Essex IG10 3TZ